System Engineering Approach to Initial Design of LEO Remote Sensing Missions

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Abstract—**The satellite remote sensing missions are essential for long-term research around the condition of the earth resources and environment. On the other hand, in recent years the application of microsatellites is of interest in many space programs for their less cost and response time. In microsatellite remote sensing missions there are tight interrelations between different requirements such as orbital altitude, revisit time, mission life and spatial resolution. Also, all of these requirements can affect the whole system level design characteristics. In this work, the remote sensing microsatellite sizing process is divided into three major design disciplines; a) orbit design, b) payload sizing and c) bus sizing. Finally, some specific design cases are investigated inside the design space for evaluating the effect of different design variables on the satellite total mass. Considering the results of the work, it is concluded that applying a systematic approach at the initial design phase of such projects provides a good insight to the not clearly seen interactions inside their highly extended design space.**

I. Introduction

The growing demand for the remote sensing missions has enabled the microsatellites to play an important role in the space technology applications. Microsatellites with their lower cost and less design time compared to the big conventional satellite projects are promising for LEO (Low Earth Orbit) remote sensing applications [1-6]. During the last decades, there has been a vast progress in the development of advanced technologies for microsatellites. These achievements resulted into the microsatellites competitive application for different space missions which remote sensing application is considerable among them [7-9]. Classically the design process of remote sensing missions starts with the definition of the mission requirements such as *RT* (Revisit Time) and *GSD* (Ground Sample Distance). On the other hand, such mission requirements are interrelated with the payload design variables like payload aperture, D_{PL}. Also, orbital characteristics like orbit altitude, h, influence the satellite subsystems design [10-13]. These interactions between the different design variables sometimes are not understandable directly and cannot be evaluated easily in the design space. To this end, developing an integrated sizing tool for different design disciplines is very useful for preliminary design trade-off studies [14]. In this work using a system engineering approach first the orbital characteristics of a practical range of SS-Os (Sun synchronous Orbits) with different altitudes and revisit times are determined. Then, according to the mission requirements, the payload sizing process starts. Afterwards and based on a reference remote sensing payload and required aperture, the mass and power budgets of payload are calculated. The mass and power budgets of other subsystems including Attitude Determination and Control, Command and Data Handling, Telecommunication, Thermal Control and Structure are calculated using design estimation relationships from [14]. Among all the satellite subsystems, the power subsystem is highly affected by the mission requirements and orbital characteristics. In order to consider the interrelations between these requirements and the power subsystem design process, a mission scenario is selected. In this scenario the remote sensing mission is scheduled to provide a global coverage belt parallel to the earth equatorial after each revisit time. After developing different design sizing tools for orbit, payload and bus, all of them are combined in a unified code in MATLAB. Some case studies are presented in order to evaluate the design space resulted from the developed sizing tool.

II. Satellite Design

Satellite design at any class and with any mission is a complex and iterative process which involves multidisciplinary engineering expertise. The design process of the remote sensing microsatellites is highly dependent on the mission requirements such as *RT* and *GSD*. These mission requirements are influenced by the type and the characteristics of the satellite orbit. Owing to this fact, orbit design variables play an important role on the whole design properties. In the following sections, the satellite sizing process is divided into

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the three major design disciplines; a) orbit design, b) payload sizing and c) bus sizing. Also, the satellite geometrical configuration is assumed to be cubic shaped with four body-mounted solar arrays.

A. Orbit design

In LEO microsatellites with remote sensing missions, SS-O is the most commonly used orbit type [5-6][9]. SS-Os are orbits with the secular rate of RAAN (Right Ascension of Ascending Node) equal to the right ascension rate of the mean sun. In this case, the position of the line of nodes remains almost the same with respect to the sun's direction [15]. This is the base of peculiar properties of SS-Os in order to achieve key remote sensing mission requirements such as providing similar lighting conditions along the satellite ground tracks throughout the mission [16]. Almost in any textbook on the satellite and its related subjects such as [11-13], there is some basic theory associated with how an orbit plane is perturbed as a result of the earth's equatorial bulge. Boain, R. J. in [17] describes the step by step process for SS-O design. Also in his work, it is explained in detailed which characteristics make SS-Os attractive for the earth remote sensing missions. In [17], it is demonstrated that *RT* (Revisit Time) can be a parameter in order to create unique SS-Os using (1):

$$T_O = 86400\ RT/R \qquad (1)$$

in which, T_O is the orbit period and R is the number of full revolutions of the satellite around the earth during each *RT*. After calculating orbit period, by applying Kepler's equation, the orbit altitudes for a SS-O can be calculated. Unquestionably, the orbit altitude has a great influence on the mission requirements and the satellite performance. Due to this fact selecting a practical range of altitudes at preliminary stages of the design is important in order to obtain meaningful trade-off studies. According to [11] the orbits with altitudes below 1000 km are considered as LEO. This upper limit is selected based on the great amount of the Van Allen radiation exposing on the satellite in higher altitudes. On the other hand, at the lower end of the altitude range atmospheric drag is the parameter which plays an important role. For orbital altitudes lower than 500 km the satellite can be affected seriously by atmospheric drag. This drag force results to slow decrement of the satellite altitude and has a negative effect on mission performance. Due to these facts, the range of altitude between 500-1000 km is selected for orbit design. In addition, this range of altitude is inside the performance margin of the majority of commercial launch vehicles [16][18].

After selecting an appropriate range for altitudes, a practical range for revisit time variation should be considered for the design process. In these projects usually this requirement is dictated by the scientific group of the project which mainly depends on the remote sensing applications. For the present sizing tool considering the practical remote sensing missions [19], the range of 3-26 days is selected for *RT* variation.

Using Boain, R. J. [17] design steps and considering the above mentioned range for the altitude and revisit time, the SS-O design is realized in the sizing tool. In Fig. 1, the revisit time is shown for different SS-Os with altitudes in range of 500-1000 km and *RT* between 3 to 10 days in which the number of revolutions is stated beside the corresponding point in each SS-O choice. For example in case of *RT*=5 days, it is seen that there are six distinct SS-O scenarios with 76, 74, 73, 72, 71 and 69 revolutions. These scenarios correspond to orbital altitude of 506, 629, 693, 758, 825 and 964 km respectively.

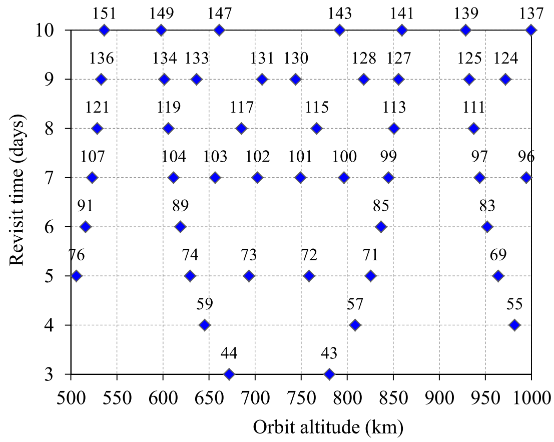

Figure 1. Revisit time versus altitude for sun-synchronous orbits.

B. Payload sizing

In the remote sensing missions, one of the important parameters in the payload design is the required *GSD* for sampling the target area. The required *GSD* depends on both the technological level of the payload instrument and the orbit altitude. Hereafter, *GSD* refers to the achievable spatial resolution which depends on optical rules and orbit altitude rather than technological constraints. According to *Rayleigh* diffraction criteria [11], the payload aperture, D_{PL}, and the angular resolution, a_r, are related with (2):

$$a_r = 1.22\lambda / D_{PL} \qquad (2)$$

in which λ is the wave length of the electromagnetic spectrum selected for the payload. On the other hand, *GSD* at the satellite nadir point is a function of the orbit altitude, h, and the angular resolution, a_r, which is calculated as:

$$\mathrm{GSD} = 2ha_r \qquad (3)$$

Therefore, *GSD* as a mission requirement is interrelated with both satellite orbit and the payload characteristics.

The electromagnetic spectrum wave length, λ, is a variable that is dictated by the science mission team in accordance with remote sensing required application. In this study, the Multi-Spectral Mid-IR remote sensing instrument with λ=4 μm is considered as the reference payload [11]. In order to calculate the mass and power of the payload, a design relationship is used as a function of $\alpha = D_{PL}/D_{Ref}$ which is the aperture ratio between the under-design payload and the reference payload. Different steps in payload sizing process are summarized in Fig. 2.

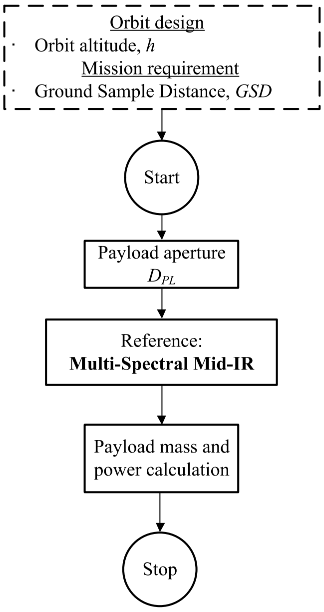

Figure 2. Payload sizing process.

C. *Bus design*

The mass and power budgets of different subsystems including Attitude Determination and Control, Command and Data Handling, Telecommunication, Thermal Control and Structure are determined based on design estimation relationships which are indicated in Table 1.

Table 1. Satellite subsystems mass and power budgets [14].

Subsystem	Mass (kg)	Power (W)
Attitude Determination and Control	-0.0142 m_{sat}+13.748	Active: 0 .0036 P_{av}+18.304 Passive: -0.0152 P_{av}+8.858
Command and Data handling	-0.0079 m_{sat} +5.5627	-0.03 P_{av}+15.39
Telecommunication	-0.0103 m_{sat} +6.5935	0.0456 P_{av}+25.583
Thermal Control	0.0498 m_{sat} +0.4785	0.0067 P_{av}+0.7862
Structure	-0.01 m_{sat} +31.079	-

The power subsystem is highly affected by the mission requirements and orbital variables. Due to this fact its design process is done considering solar array sizing and battery sizing in the coming sections. This approach gives a better understanding of interrelation between mission requirements and power subsystem variables which affect the whole system design process. In the power subsystem sizing process the classical formulas are used from different resources [11-13].

- *Solar array sizing*

Solar array design process is mainly affected by the power consumption profile of the different subsystems and remote sensing payload. Power required to be produced by the solar arrays is calculated by (4):

$$P_{sa}=(P_eT_e/X_e+P_dT_d/X_d)/T_d \quad (4)$$

in which, P_{sa} is the power required from solar arrays during daylight, P_d is the power required by satellite during orbit daylight, P_e is the power required by satellite during eclipse, T_e is the eclipse duration, T_d is the orbit daylight duration, X_e is the power transmission efficiency from solar arrays to batteries and then to individual loads (~0.60), and X_d is the power transmission efficiency from solar arrays to loads (~0.80).

Chobotov, V. A. in [20] suggest a formula for calculating the eclipse as:

$$f_e = \pi^{-1}\,(\cos^{-1}[(h^2+2R_eh)^{0.5}/(R_e+h)\cos\beta]) \quad (5)$$

where $f_e= T_e/T_O$ is the eclipse fraction of the orbit and β is the sun angle. The exact amount of β depends on launch window and usually is calculated with computer programs with algorithms for both the solar ephemeris and for propagating the orbit elements in time. For the present work applicable to early design phases, it is assumed that β=0º which implies that the sun is located in orbital plane and at 12:00 AM and 12:00 PM local times the satellite will pass above the earth equatorial. After calculating the orbit day light duration, T_d, and eclipse, T_e, in order to calculate the required power in each of these periods, the remote sensing mission scenario is assumed with the following characteristics:

- Remote sensing mission is realized during the orbit daylight as well as eclipse with equal duration in each period, t_{PL} is the whole duration of remote sensing mission.
- Mission is realized at the same latitudes in orbit daylight and eclipse in order to have a uniform coverage parallel to the earth equatorial after each revisit time.
- Subsystems operations are scheduled according to Table 2.

Table 2. Satellite subsystems operations schedule per orbit.

Subsystem	Operation schedule
Attitude Determination and Control	Active: 2 min before and 2 min after mission. Passive: in the rest of orbit.
Command and Data handling	Active through all orbit.
Telecommunication	Active during t_{COM} = 2t_{PL}.
Thermal Control	Active during eclipse.

Considering the subsystem operations schedule and the power budget of different subsystems indicated in Table 1, the total power which should be provided by solar arrays per orbit, P_{sa}, is calculated. Also, the mass of the four solar arrays is calculated based on P_{sa} and the properties of two typically used solar cells, Si (Silicon) and GaAs (Gallium-Arsenide).

➢ *Battery sizing*

Battery sizing process starts by calculating the required capacity of the batteries for the whole *ML* (Mission Life). The required capacity is calculated by:

$$C_r = P_e T_e / X_{b\text{-}l} \mathrm{DOD} \quad (6)$$

Where $X_{b\text{-}l}$ is the power transmission efficiency from batteries to loads (~0.70), and DOD is the battery depth of discharge. DOD is defined as the percent of total battery capacity removed during a discharge period. Higher percentages imply shorter *CL* (Cycle Life), as shown in Fig. 3.

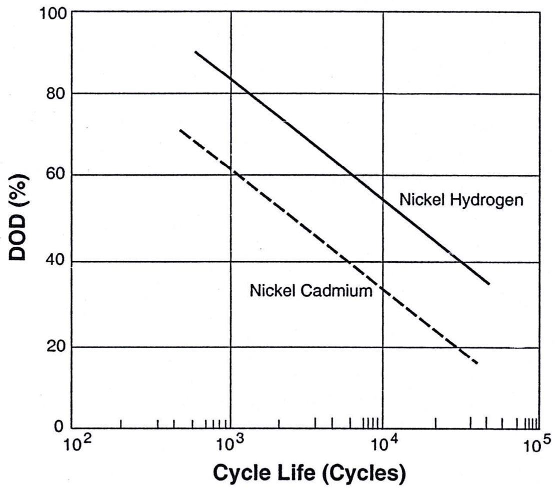

Figure 3. Depth-of-discharge versus cycle life for batteries [11].

In section *A. Orbit design*, it is described that each SS-O represents a specific number of revolutions in a specific revisit time. For example according to Fig. 1 a SS-O with 810 km altitude and *RT*= 4 days, the satellite should complete 57 revolutions. Considering this, the satellite *CL* (Cycle Life), is defined by (7):

$$CL = 365 ML \times R / RT \quad (7)$$

in which *ML* is the mission life of the satellite in years and *R* is the number of full revolutions of the satellite around the earth during *RT*. By calculating *CL* from (7), the amount of DOD is determined from Fig. 3 depending of the battery type. Finally considering the mission scenario indicated in section *D. Solar array sizing* and having the amount of the required power during eclipse, P_e, the required capacity is calculated for the batteries. The mass of the batteries is calculated based on the required capacity and the properties of the battery type. In the power subsystem sizing tool two types of batteries, NiCd (Nickel Cadmium) and NiH2 (Nickel Hydrogen) are considered as the design choices for the battery sizing. The power subsystem total mass is calculated from the mass of solar arrays and the batteries plus 30% margin to account for the power control units electronics and required harness. The different steps in power subsystem design process are shown in Fig. 4.

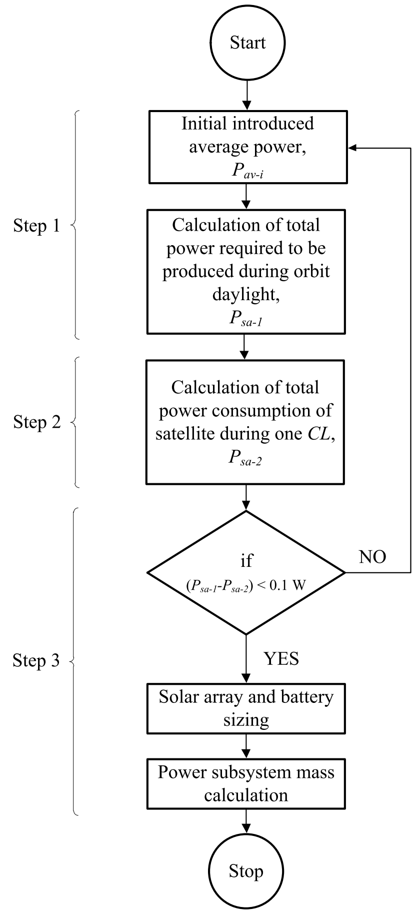

Figure 4. Power subsystem sizing steps.

III. SATELLITE SIZING TOOL

In order to evaluate the interrelations between different design variables in the design space, the three design disciplines described in previous section are combined in a unified code called satellite sizing tool hereafter. Different steps of the satellite sizing tool are seen in Fig. 5.

In step 1, the orbital characteristics for different SS-O choices are determined and input variables are set. In step 2, according to the orbit altitude and the process indicated in Fig. 2, the remote sensing payload mass and power budgets are calculated. In step 3, an initial value for the satellite total mass, m_i, is introduced and based on the design estimation relationships indicated in Table 1 the mass of all the satellite subsystems except power subsystem is calculated. In step 4, first the power subsystem mass is calculated according to the process shown in Fig. 4. Then it is added to the payload mass and other subsystems mass calculated in steps 2 and step 3 respectively. Now, the satellite total mass, m_{sat}, is calculated. At this step, if the difference between the calculated satellite total mass, m_{sat} , with the initial introduced mass, m_i, is less than 0.5 kg the sizing process stops successfully. If not m_i is modified to fulfill this condition.

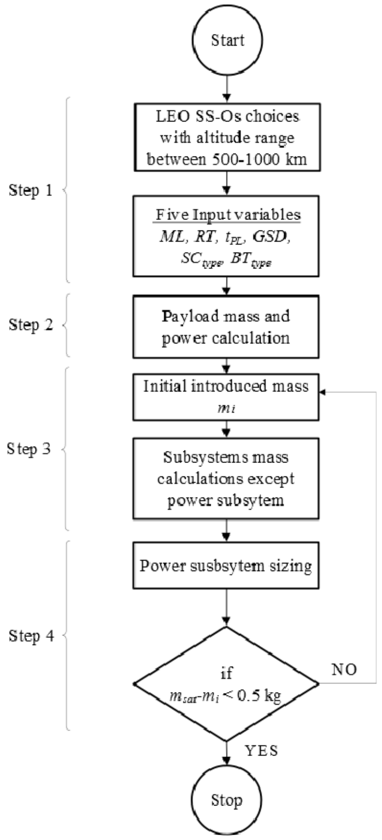

Figure 5. Satellite sizing tool calculation steps.

The developed sizing tool provides an extended design space which contains both mission and system design variables. Design variables consist of four mission variables as well as two system variables. The mission variables include *ML*, *RT*, t_{PL} and *GSD*. The system level variables are representative for technology choices in selection of solar cells between Si and GaAs and battery cells between NiCd and NiH2. The range of variation of mentioned variables are indicated in Table 3.

Table 3. Satellite sizing tool variables.

Variable	*ML* (year)	*RT* (day)	t_{PL} (min)	*GSD* (m)	SC_{type}	BT_{type}
Variation limit	[3,5]	[3-26]	[5,20]	[30,200]	Si or GaAs	NiCd or NiH2

IV. Results

Some specific design cases are investigated inside the design space for the effect of different design variables on the satellite total mass. The results are presented in Fig. 6 to Fig. 9.

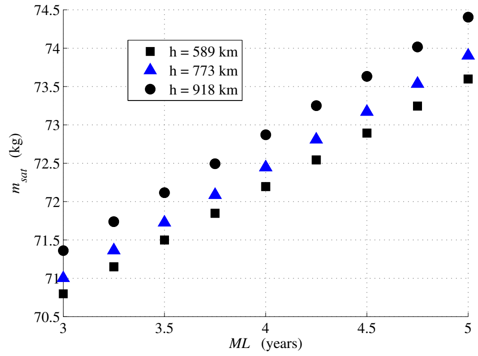

Figure 6. Mission life, *ML*, effect on satellite total mass, m_{sat}, when: *RT*=14 days, t_{PL}=10 min, *GSD*=200 m, SC_{type}=Si, BT_{type}=NiCd.

As seen in Fig. 6, by increasing the mission life the satellite total mass increases. This is due to the mass increment in the power subsystem in order to support the satellite during longer operation in orbit. In Fig. 6, all the three indicated altitudes are representatives for *RT*=14 days. The small difference between the satellite total mass with altitude variation also is because of the variation in power subsystem mass according to the changes in orbit day light period, T_d, and orbit eclipse, T_e, with orbit altitude variation.

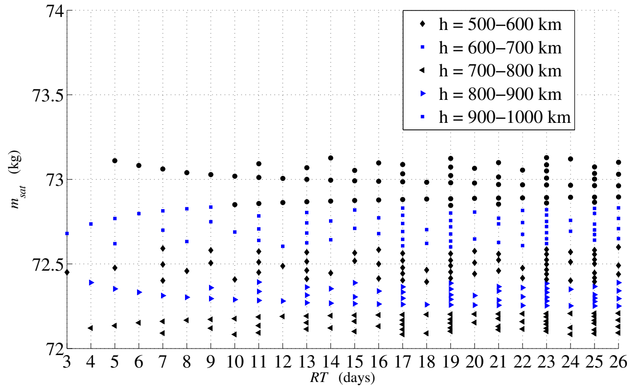

Figure 7. Revisit time, *RT*, effect on satellite total mass, m_{sat}, when: *ML*=4 years, t_{PL}=10 min, *GSD*=200 m, SC_{type}=Si, BT_{type}=NiCd.

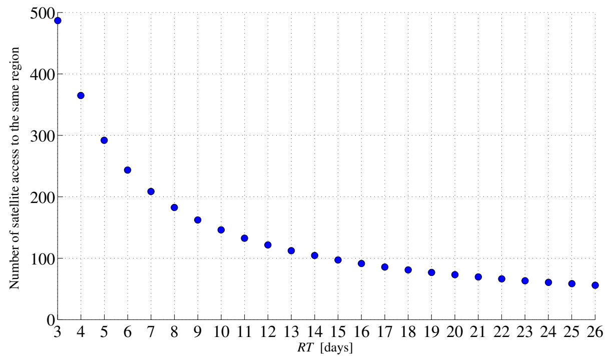

Figure 8. Revisit time, *RT*, effect on number of satellite access to the same region for *ML*=4 years.

In Fig. 7, it is seen that for different altitudes between 500-1000 km, the effect of revisit time, *RT*, on the satellite total mass is not considerable. So, it can be concluded that revisit time itself has not a major effect on a system level characteristic like the satellite total mass for a specific mission life *ML*=4 years. But as seen in Fig. 8 with the same mission life duration the satellite access to the remote sensing target area is decreasing when the revisit time increases. This result gives useful insight to the design team for appropriate decision making considering the interrelation between a system level characteristic, satellite total mass, and a mission performance variable as number of satellite access to the target area.

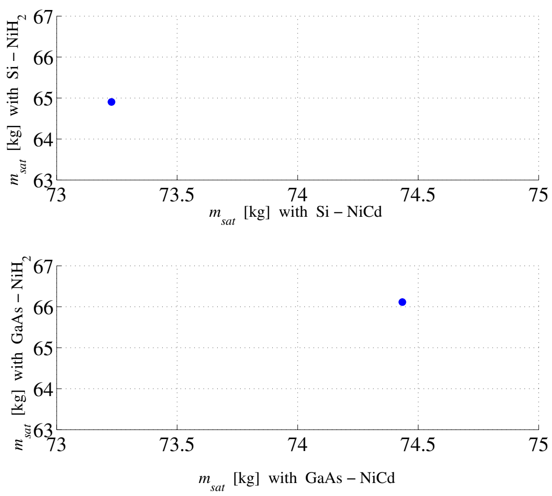

Figure 9. Solar cell and battery type effect on satellite total mass, m_{sat}, when: *ML*=4 years, *RT*=14 days, t_{PL}=10 min, *GSD*=100 m.

Fig. 9 shows the satellite total mass changes according to the technology choices for solar cells and battery types. According to Fig. 9, among all of the four possible combinations of different choices, the Si-NiH2 combination results to the lower satellite total mass for a concrete design point.

V. CONCLUSIONS

Satellite design is among of highly coupled system design problems. It is a process in which the mission requirements and orbital parameters have direct and indirect correlations with system design characteristics. Due to this fact applying a systematic approach at the early phases of the design is useful for design decision makers. In this work, focusing on the SS-O remote sensing missions, the interactions between the three different design disciplines as orbit, payload and bus design are investigated. The selected design variables consist of four mission variables as well as two system variables. The mission variables include mission life, *ML*, revisit time, *RT*, remote sensing mission duration, t_{PL}, and *GSD*. The system level variables are selected as different technology choices for solar cells between Si and GaAs and battery cells between NiCd and NiH2. Some case studies are presented inside the extended design space produced by the sizing tool. The obtained results help the design team to understand the interrelation between mission requirements and system design characteristics.

VI. FUTURE WORKS

Different mission scenarios and more technology choices for power subsystem can be added to the design sizing tool. Verification of results based on real remote sensing microsatellites. Also, the developed sizing tool can be linked to appropriate optimization algorithms for looking for optimum design. This can be performed in the context of MDO (Multidisciplinary Design Optimization) methodology.

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